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EXPERIMENTAL STUDY OF EFFECTS OF GEOMETRIC VARIABLES ON PERFORMANCE OF CONTOURED ROCKET-ENGINE EXHAUST NOZZLES

By Harry E. Bloomer, Robert J. Antl, and Paul E. Renas

Lewis Research Center
Cleveland, Ohio

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E-1403

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EXPERIMENTAL STUDY OF EFFECTS OF GEOMETRIC VARIABLES ON PERFORMANCE OF CONTOURED ROCKET-ENGINE EXHAUST NOZZLES

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SUMMARY

An investigation was conducted to determine the effect of exhaust nozzle contouring on performance and separation characteristics. The nozzles were mounted on a water-cooled JP-4 fuel - gaseous-oxygen thrust chamber with a 2.67-inch-diameter throat. The range of variables included nozzle area ratios of 16, 25, and 30 and nozzle pressure ratios from 35 to 450.

The results of the investigation, when compared with results obtained in a previous investigation with conical nozzles, showed that the contoured nozzles with area ratios of 25 and 30 had vacuum thrust coefficients 1 to 2 percent higher than optimum conical nozzles of the same length. For nozzles of the same length and area ratio, a performance gain of 1 percent at design pressure ratio was realized by contouring. Contoured-nozzle separation data agreed with a correlation based on the Mach number ratio across the oblique shock wave which occurs at the separation point.

INTRODUCTION

Rocket-engine performance is markedly affected by nozzle pressure ratio, or altitude. Ground-launched systems are optimized for a particular design altitude; overexpansion losses are suffered at lower altitudes, and underexpansion losses are encountered at higher altitudes. The importance of nozzle efficiency in upper stages is demonstrated in a typical 24-hour satellite mission, where a 1-percent increase would allow the payload to be increased approximately 7 percent. It is necessary to know the efficiency in order to predict mission parameters for a particular system and, of course, to optimize the design of new systems in terms of nozzle performance and weight. One approach has been to contour the nozzle walls in order to obtain both large exit- to throat-area ratio and near-axial-flow discharge angles in a limited length.

Investigations to determine effects of geometric variables on performance of rocket-engine exhaust nozzles were undertaken at the NASA Lewis Research Center to augment the many experimental and analytical studies that are reported in references 1 to 9. Studies of the effects of divergence angle and area ratio on conical nozzle performance and separation characteristics are reported in reference 10. The studies reported herein show the effect of nozzle contouring on nozzle performance and separation characteristics. The range of variables included nozzle area ratios of 16, 25, and 30 and nozzle pressure ratios from 35 to 450. A thrust-chamber design that would be fairly durable and of modest scale, utilize common propellants, and be restartable was desired for these studies. A water-cooled JP-4 fuel - gaseous-oxygen thrust chamber with a 2.67-inch-diameter throat was selected for the investigation. The chamber was operated over a range of propellant mixtures from 23 to 33 percent fuel.

E-1403

The investigation was conducted in an altitude facility that permitted operation of the thrust chamber at a constant chamber pressure of approximately 330 pounds per square inch and nozzle exit pressures as low as 0.25 pound per square inch absolute. The results are presented in tables and in graphical form to show the trends of nozzle performance parameters as functions of nozzle design and operating variables. These results are also compared with the conical nozzle performance reported in reference 10.

SYMBOLS

A area, sq in.

C_F thrust coefficient; $C_F = \frac{Ig}{c^*} = \frac{F}{P_c A_t}$

c effective exhaust velocity, $c^* C_F$, ft/sec

c^* characteristic exhaust velocity, $gP_c A_t / W_p$, ft/sec

D diameter, in.

F thrust, lb

g acceleration due to gravity, 32.174 ft/sec^2

I specific impulse, F/W_p, (lb force)(sec)/lb mass

L length, in.

o/f oxidant-fuel weight ratio

P total pressure, lb/sq in.

p static pressure, lb/sq in.

r radius, in.

T temperature, °R

W mass-flow rate, lb/sec

α nozzle divergence half-angle, deg

ϵ ratio of nozzle exit to throat area, A_e/A_t

θ wall slope angle, deg

Subscripts:

c combustion chamber

e nozzle exit

eq equilibrium

f fuel

n nozzle surface

p propellant

ox oxidant

sep separation

t nozzle throat

t' nozzle throat divergent portion

X nozzle axial station

0 ambient conditions

APPARATUS

Rocket Thrust Chamber Configurations

A schematic drawing of the rocket thrust chamber is presented in figure 1. The portion upstream of the throat (combustion chamber) was the same for all nozzle configurations. Three contoured-nozzle configurations having area ratios of 16, 25, and 30 were used in this investigation. The contoured-nozzle coordinates were obtained by the method of reference 11. The length of the contoured nozzle with an area ratio of 16 was equivalent to 80 percent of the length of a 15° half-angle conical nozzle having the same area ratio. The lengths of the contoured nozzles with area ratios of 25 and 30 were equivalent to 60 percent of the length of 15° half-angle conical nozzles having the same respective area ratios. Detailed dimensions of the three nozzle configurations are presented in table I.

The injector used for the tests consisted of alternate concentric rings of like-on-like fuel orifices and showerhead oxidant orifices, with the exception of the outer ring, which was like-on-like fuel orifices alternating with showerhead fuel orifices. Fuel was supplied to the injector by means of high-pressure pumps. The gaseous oxygen was supplied from a number of containers that were manifolded together and initially pressurized to 2300 pounds per square inch. Pilot flows of gaseous hydrogen and oxygen introduced through the injector were ignited by a sparkplug mounted in the center of the injector. The entire thrust chamber was cooled with water. The water manifold at the exit of the nozzle was supplied with about 110 gallons per minute of water at high pressure. Water leaving the thrust chamber at the injector end was drained to a sewer.

The combustion chamber and the injector are the same as those used for the studies reported in reference 10.

Facility

The rocket thrust chamber was installed in an altitude test chamber where altitude pressures were maintained by the facility exhaust system. Sufficient air to burn residual combustibles in the exhaust gas safely was supplied through the facility combustion air system. Data for the required amount of air were obtained from studies reported in reference 12. A water spray was used to cool and further inert the exhaust gases after they were burned.

Photographs of the rocket thrust chamber, the injector, and the test stand are presented in reference 10.

Instrumentation

Thrust measurements were obtained by use of a strain-gage transducer mounted on the test stand and were transmitted to an automatic data recorder and to a direct-writing oscillograph in the control room.

Fuel flow measurements were made with a vane-type flowmeter. These measurements were transmitted to a read-out counter in the control room, an automatic data recorder, and an oscillograph. Oxygen flow measurements were made with a sharp-edged orifice. The total pressure upstream of the orifice and the pressure drop across it were measured with pressure transducers and the results transmitted to the automatic data recorder and to the oscillograph. The temperature of the oxygen upstream of the orifice was measured with an iron-constantan thermocouple and was recorded on the automatic data recorder.

Combustion-chamber pressure was measured with a pressure transducer. These measurements were recorded on the automatic data recorder and the oscillograph and also shown on a gage located in the control room.

The nozzle was instrumented with wall taps at points along the inner surface to measure the static-pressure distribution. Static-pressure instrumentation was also located on the nozzle lip to measure ambient pressure. These measurements were transmitted from pressure transducers to the automatic data recorder and to the oscillograph.

PROCEDURE

After proper altitude pressure, bypass airflow, and fuel and oxidant settings were made, a time sequencing system was employed to control automatically the events of the rocket firing.

Pilot flows of gaseous hydrogen and oxygen were introduced through the injector and ignited by the sparkplug located in the center of the injector. The main propellants were then introduced and the pilot flows were stopped. The automatic data recorder and the oscillograph were started approximately 1 second before the rocket firing. The rocket was allowed to run for approximately 30 seconds to ensure that all parameters were at steady conditions. Performance data presented herein are based on data recorded during the last second of running.

RESULTS AND DISCUSSION

The results of the investigation are described in the following order: (1) performance data for the contoured nozzles are presented and discussed; (2) nozzle performance data for contoured and conical nozzles are compared over a range of pressure ratios, extrapolated to vacuum conditions, and then shown at design pressure ratio; (3) nozzle separation characteristics are discussed.

The contoured nozzles are referred to as a percent of the length of a 15° half-angle conical nozzle having the same area ratio. Hereafter, the contoured nozzle with an area ratio of 16 will be referred to as an 80-percent-length nozzle and the contoured nozzles with area ratios of 25 and 30 as 60-percent-length nozzles.

Nozzle Performance

Performance data for contoured-nozzle configurations. - Performance parameters for the contoured nozzles are presented in table II. Nozzle thrust coefficient and specific impulse from these tabulations are shown graphically as functions of pressure ratio p_0/P_c at two oxidant-fuel ratios in figures 2 to 4. The variation of nozzle thrust coefficient with nozzle pressure ratio for the nozzles with area ratios of 16 and 25 (figs. 2(a) and 3(a)) is linear over the entire range of nozzle pressure ratios tested, indicating full nozzle flow. The curve for the nozzle with an area ratio of 30 (fig. 4(a)) is linear only at pressure ratios less than about 0.016, which indicates that separation occurred at pressure ratios above this value. Specific impulse (figs. 2(b), 3(b), and 4(b)), as would be expected, shows the same trends as thrust coefficient.

The effect of mixture ratio on thrust coefficient and specific impulse was small. The higher ratio (2.85) gave higher values of thrust coefficient and specific impulse. Examination of the data shows that the effect of mixture ratio on specific impulse results directly from its influence on thrust coefficient rather than from a variation in characteristic exhaust velocity c^*. The data in table II show that the average c^*, over the range of data accumulated, varied between 5625 and 5725 feet per second, 5675 being the mean. The mean characteristic exhaust velocity with 29 percent fuel in the propellant (oxidant-fuel ratio of 2.45) is about 96 percent of theoretical equilibrium c^*, while with 26 percent fuel in the propellant (oxidant-fuel ratio of 2.85) the mean c^* is approximately 98 percent of the theoretical value.

A discussion of the accuracy of the parameters used in calculations of thrust coefficient, specific impulse, and characteristic exhaust velocity is presented in reference 10. As a further aid in the discussion of accuracy, a comparison of thrust coefficients calculated from thrust and from nozzle pressure measurements is also presented therein.

Contoured-nozzle performance comparison. - Performance data obtained with contoured nozzles having area ratios of 16, 25, and 30 are compared in figure 5. Faired curves showing the variation of nozzle thrust coefficient C_F with nozzle pressure ratio P_c/p_0 and altitude are presented. The altitude scale is based on a thrust-chamber pressure of 600 pounds per square inch. These data illustrate nozzle performance trends over the range of pressure ratios that would be expected during a boost trajectory. The lowest area ratio nozzle shows the best performance at pressure ratios up to 300 (altitudes below 46,000 ft). At pressure ratios above 300, the larger area ratio nozzles are superior.

A comparison of the performance of the contoured nozzle with an area ratio of 30 (60 percent length) with that of conical nozzles (data from ref. 10) having the same nominal area ratio is shown in figure 6. An

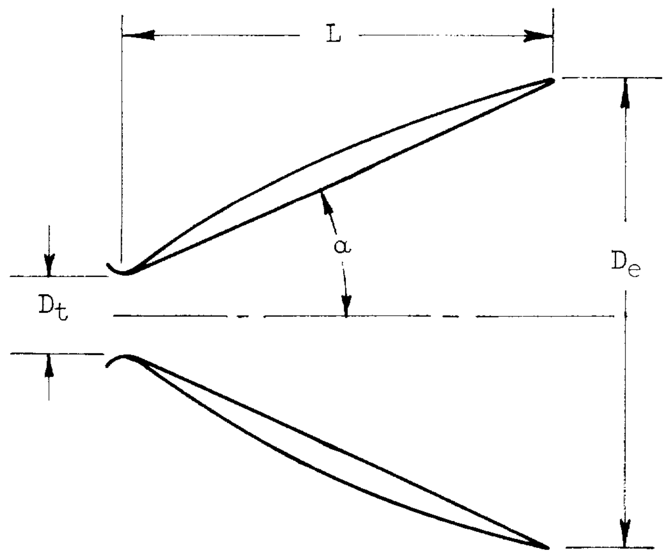
E-1403

altitude scale based on a thrust-chamber pressure of 600 pounds per square inch is also shown in the figure. At a nozzle pressure ratio of 40 (sea level), the contoured nozzle had a much lower thrust coefficient than the conical nozzles because of poorer separation characteristics. Similar results have been observed in past experiments with contoured nozzles. At pressure ratios above 100 (altitudes above 22,000 ft) the contoured nozzles followed the performance trends of the conical nozzles. This is to be expected, since all nozzles are flowing full at the higher pressure ratios and the only increase in thrust coefficient is due to the underexpansion term, which is the same for all nozzles of the same area ratio at any specified pressure ratio. The level of performance of the contoured nozzle in this region is about equal to that of a 22.5° conical nozzle.

Effects of nozzle length and weight parameters on performance in a vacuum. - In figure 7, the performance of the contoured nozzles is compared with the performance of conical nozzles (data of ref. 10) on the basis of nozzle length and weight parameters. The weight parameter is based on total surface area of the nozzle, which has been considered in other investigations (e.g., ref. 13) to be closely related to the actual nozzle weight for area ratios less than 30. The length and weight parameters have been made nondimensional by dividing length and surface area by the throat radius and area, respectively, and thus the data may be applied to larger scale nozzles. The comparison shown in figure 7 is for 29 percent fuel in the propellant; the trends for 26 percent fuel in the propellant were similar.

For equal length nozzles, the data (fig. 7(a)) indicate that contouring can give performance advantages of 1 to 2 percent over straight-wall nozzles. For nozzles of equal weight, the data (fig. 7(b)) indicate that contouring offers no advantage over the best straight-wall nozzle. Data obtained with the contoured nozzle fell approximately on the curve for the 20° half-angle conical nozzle.

Comparison of nozzle thrust coefficients and discussion of losses at design pressure ratio. - The performance gains realized by contouring are further illustrated by comparison at design pressure ratio of the percent theoretical equilibrium thrust coefficient of contoured and conical nozzles having the same length and area ratio. The contoured nozzle with an area ratio of 16 is equivalent to an 18.5° half-angle conical nozzle, and the contoured nozzles with area ratios of 25 and 30 are equivalent to 24° half-angle conical nozzles. (The following sketch shows a contoured nozzle and the equivalent conical nozzle.) An attempt was made to separate the various losses of the contoured nozzles and their equivalent conicals. As shown in the following table, the calculated divergence losses (based on the leaving angle of the nozzles) are smaller for the contoured nozzles than the equivalent conicals. However, the "other losses" for the contoured nozzles (e.g., condensation, formation of solid products during expansion, incomplete chemical recombination) are much larger. Since the nozzles are so similar geometrically, the "other losses" would not be expected to vary this much. Therefore, the conclusion is that the actual divergence losses for the contoured nozzles are probably greater than the calculated values. The data indicate that net gains of 0.7 to 1.5 percent in thrust coefficient at design pressure ratio can be realized by applying the contouring method of reference 11.



Area ratio, ϵ	Nozzle	Percent of $C_{F,eq}$	Divergence losses, percent	Losses upstream of throat, percent	Viscous losses, percent	Other losses, percent
16	80-percent-length contoured	96.0	0.7	0.6	0.3	2.4
16	18.5° half-angle conical	95.3	2.6	.6	.4	1.1
25	60-percent-length contoured	95.0	1.3	.6	.2	2.9
25	24° half-angle conical	93.5	4.3	.6	.3	1.3
30	60-percent-length contoured	94.2	1.3	.6	.2	3.7
30	24° half-angle conical	93.3	4.3	.6	.3	1.5

E-1403

Nozzle Flow Characteristics

Nozzle pressure profiles over the range of ambient pressures tested are presented in figure 8 for 29 percent fuel in the propellant (oxidant-fuel ratio of 2.45). The theoretical equilibrium and frozen expansion curves are included for comparison. Also shown, for comparative purposes, is an experimental full-flow curve for a 20° half-angle conical nozzle from reference 10. The overall nozzle pressure ratio values are listed in the figure. At area ratios above 5, the contoured-nozzle full-flow expansion curves lie above the theoretical equilibrium and frozen curves and also above the experimental conical nozzle data. This is typical of contoured-nozzle designs, where initial large flow expansion angles after the throat require large compressive turning in short nozzle lengths. Of course, no compressive turning is achieved in a conical nozzle, and this accounts for the large disparity in nozzle wall pressures.

The pressure profile for the contoured nozzle with an area ratio of 30 (fig. 8(c)) shows two cases of nozzle separation. These points agreed with the generalization of nozzle separation characteristics applied to (hot flow) rocket nozzles in reference 10. This generalization was made by use of the Mach number ratio across the oblique shock wave that occurs at the separation point. The Mach number ratio was approximately 0.85 over the range of nozzle pressure ratios investigated for a specific-heat ratio of 1.2.

SUMMARY OF RESULTS

An investigation of the performance and flow separation characteristics of a family of contoured rocket exhaust nozzles with area ratios of 16, 25, and 30 was conducted over a wide range of nozzle pressure ratios. The data, when compared with similar data previously obtained with a family of conical rocket exhaust nozzles, yielded the following results:

1. For a given area ratio, and at pressure ratios above 100, the contoured-nozzle performance was about equal to that of a 22.5° half-angle conical nozzle.

2. For a given length, the contoured nozzles designed to have a length of 60 percent of an equivalent-area-ratio 15° half-angle conical nozzle had vacuum thrust coefficients from 1 to 2 percent higher than the best conical nozzle (25° half-angle). The contoured nozzle designed to have a length of 80 percent of an equal-area-ratio 15° half-angle conical nozzle indicated no advantage over the best conical nozzle at vacuum conditions. For a given surface area (an indication of nozzle weight), the contoured nozzles had vacuum thrust coefficients approximately equal to the best conical nozzle (20° half-angle).

3. For the same nozzle length and area ratio, a gain of approximately 1 percent in design-pressure-ratio thrust coefficient can be expected from contouring.

4. Nozzle separation data agreed with a correlation based on the Mach number ratio across the oblique shock wave which occurs at the separation point. The correlation was successfully applied to (hot flow) rocket nozzles in a previous investigation.

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio, October 10, 1961

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E-1403

TABLE I. - DIMENSIONS OF CONTOURED-NOZZLE CONFIGURATIONS

(a) Thrust-chamber and nozzle parameters

Nominal area ratio, ϵ	Average throat diameter, D_t, av, in.	Nozzle axial length (throat to exit), L, in.	Ratio of nozzle length to throat radius, L/r_t	Throat area, A_t, sq. in.	Nozzle exit diameter, D_e, in.	Nozzle exit area, A_e, sq. in.	Exact area ratio, A_e/A_t	Nozzle surface area, A_n, sq. in.	Ratio of nozzle surface area to throat area, A_n/A_t	Wall slope angle, deg: Throat divergent portion, θ_t'	Wall slope angle, deg: Nozzle exit, θ_e	Equivalent length of 15° half-angle conical, percent
16	2.663	12.02	9.03	5.57	10.71	90.11	16.18	295.5	53.05	29.38	9.50	80
25	2.650	12.38	9.35	5.52	13.32	139.35	25.24	373	67.57	35.80	13.17	60
30	2.668	13.64	10.22	5.59	14.78	171.45	30.67	445	79.61	35.80	13.17	60
1	2.670	-----	-----	5.60	2.670	5.60	1	-----	-----	------	------	--
1	2.674	-----	-----	5.61	2.674	5.61	1	-----	-----	------	------	--

(b) Nozzle coordinates (see fig. 1)

Nominal area ratio of 16		Nominal area ratio of 25		Nominal area ratio of 30	
X, in.	Y, in.	X, in.	Y, in.	X, in.	Y, in.
0.258	1.403	0.385	1.459	0.403	1.474
.668	1.630	.521	1.565	1.135	2.013
1.335	1.989	.798	1.749	2.095	2.690
2.003	2.328	1.085	1.948	2.763	3.121
2.670	2.650	1.368	2.143	3.360	3.471
3.338	2.952	1.661	2.339	4.086	3.896
4.005	3.233	2.242	2.714	5.967	4.826
4.673	3.495	2.821	3.066	7.195	5.353
5.340	3.735	3.400	3.400	9.010	6.034
6.008	3.958	3.985	3.718	10.480	6.513
6.675	4.163	4.574	4.017	13.640	7.390
7.343	4.352	5.140	4.287		
8.010	4.526	5.713	4.539		
8.678	4.686	6.531	4.882		
9.345	4.835	7.655	5.311		
10.013	4.974	8.851	5.720		
10.680	5.104	10.129	6.109		
11.348	5.225	11.493	6.471		
12.015	5.355	12.383	6.660		

E-1403

TABLE II. - ROCKET PERFORMANCE DATA

Area ratio, ε	Thrust, F, lb	Chamber pressure, P_c, lb/sq in.	Ambient pressure, p_0, lb/sq in.	Nozzle pressure ratio, P_c/p_0	Nozzle pressure ratio, p_0/P_c	Nozzle thrust coefficient, C_F	Fuel flow, w_f, lb/sec	Oxidant flow, w_{ox}, lb/sec	Propellant flow, w_p, lb/sec	Fuel in propellant, percent	Oxidant-fuel ratio, o/f	Effective exhaust velocity, c, ft/sec	Characteristic velocity, c^*, ft/sec	Specific impulse, I, sec
14	2140	[illegible]	1.02	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
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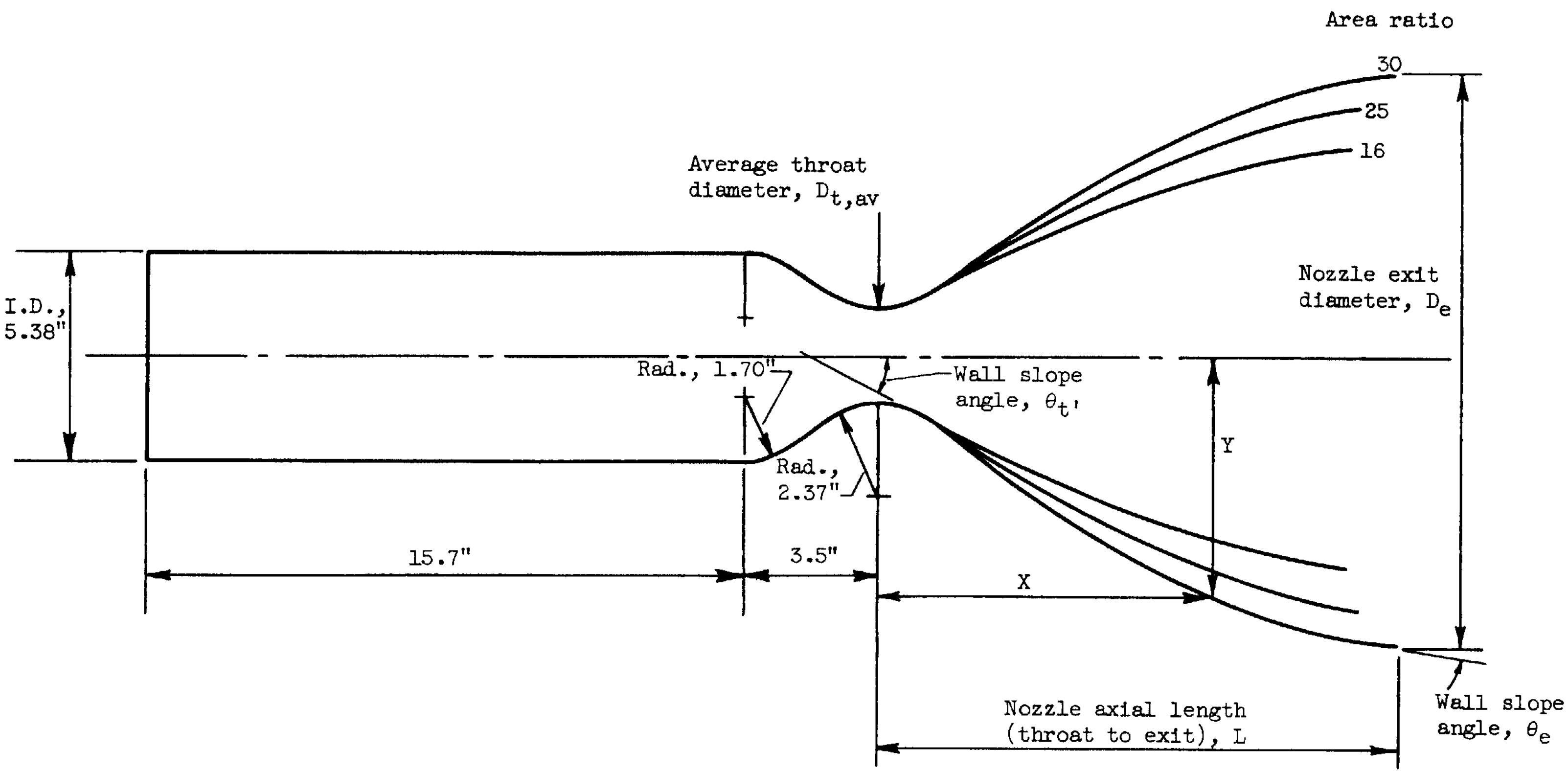
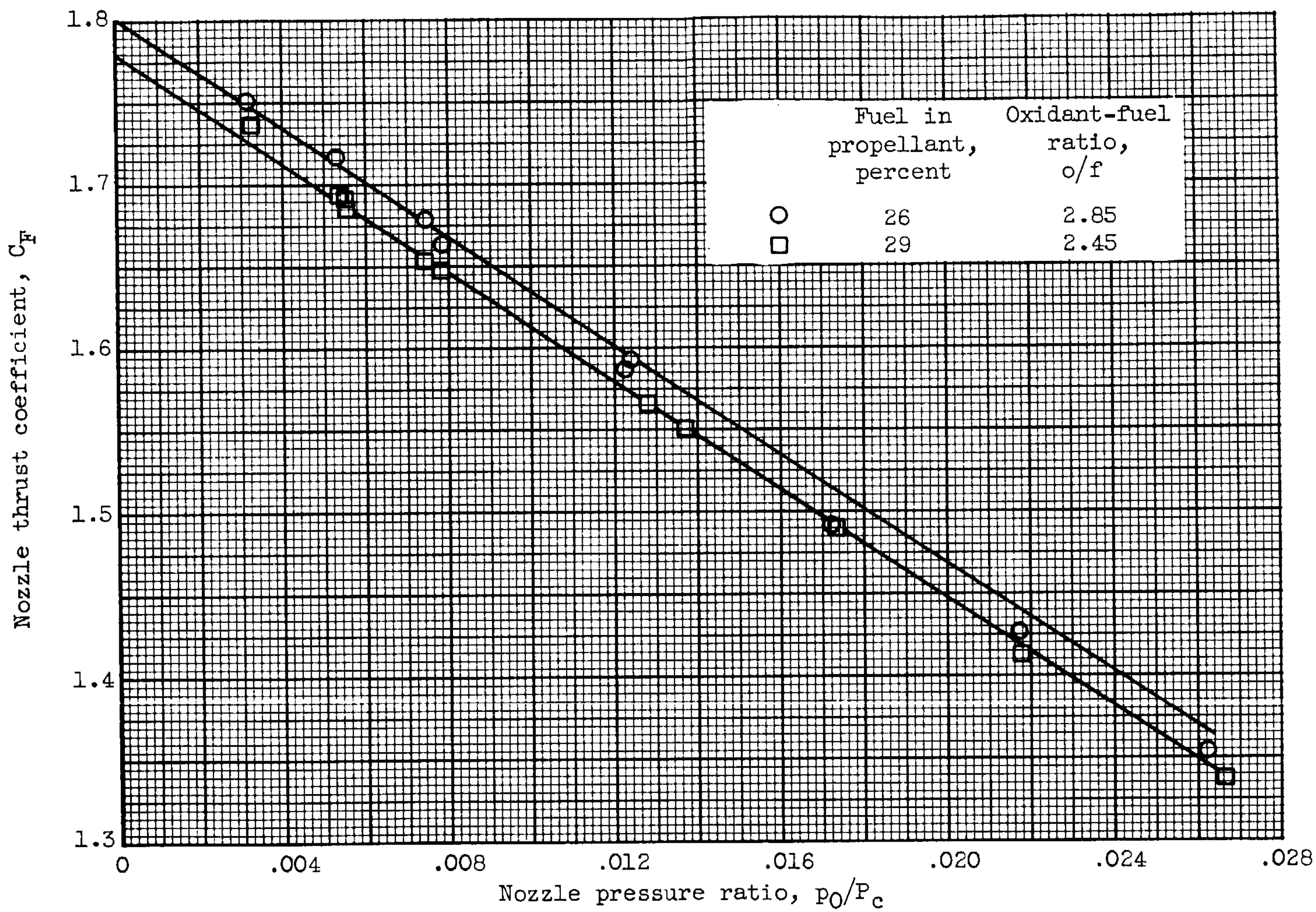
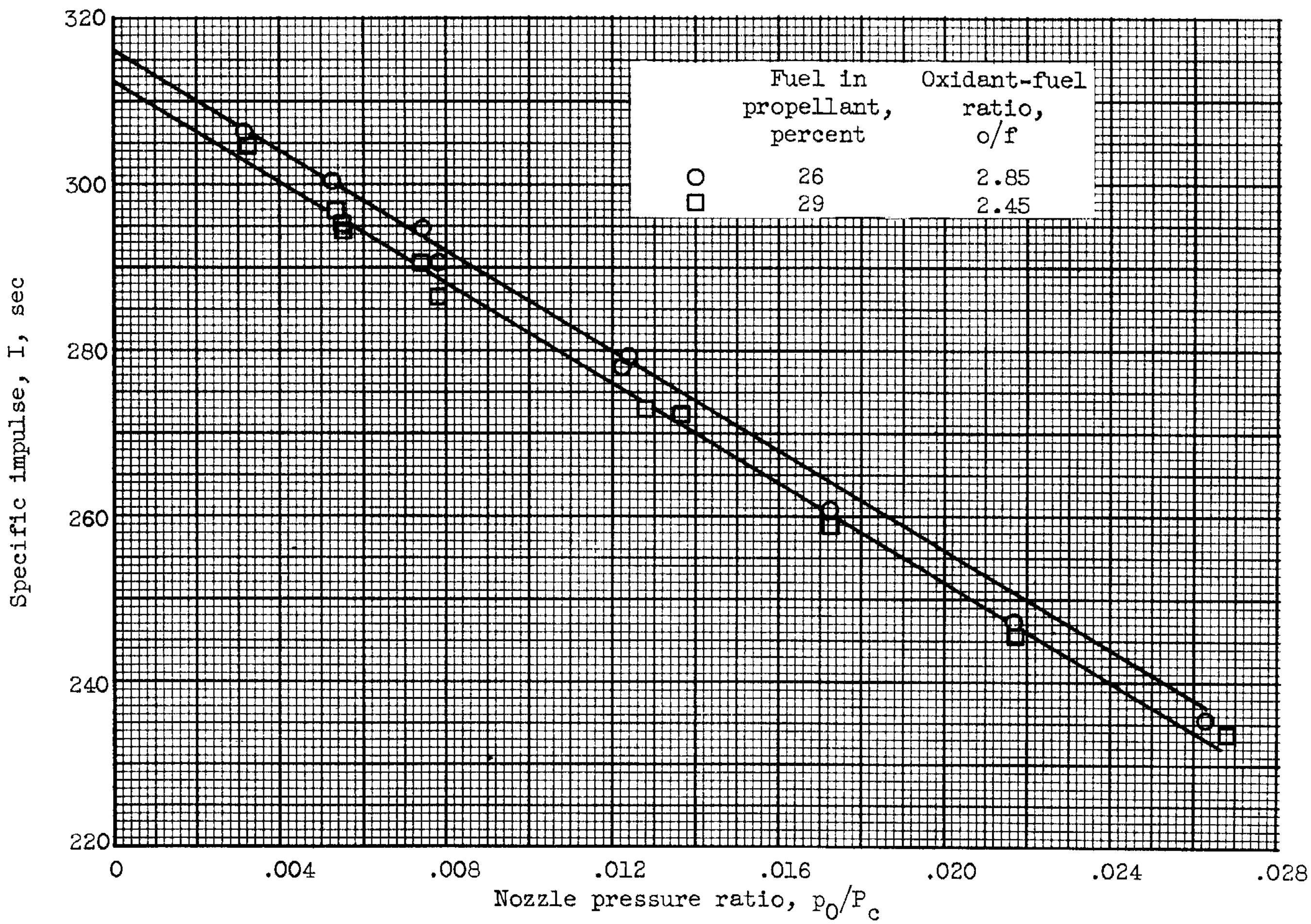


Figure 1. - Schematic detail of rocket thrust chamber and nozzles. (Exact dimensions for D_t, D_e, L, and nozzle coordinates X and Y can be found in table I.)



(a) Nozzle thrust coefficient.

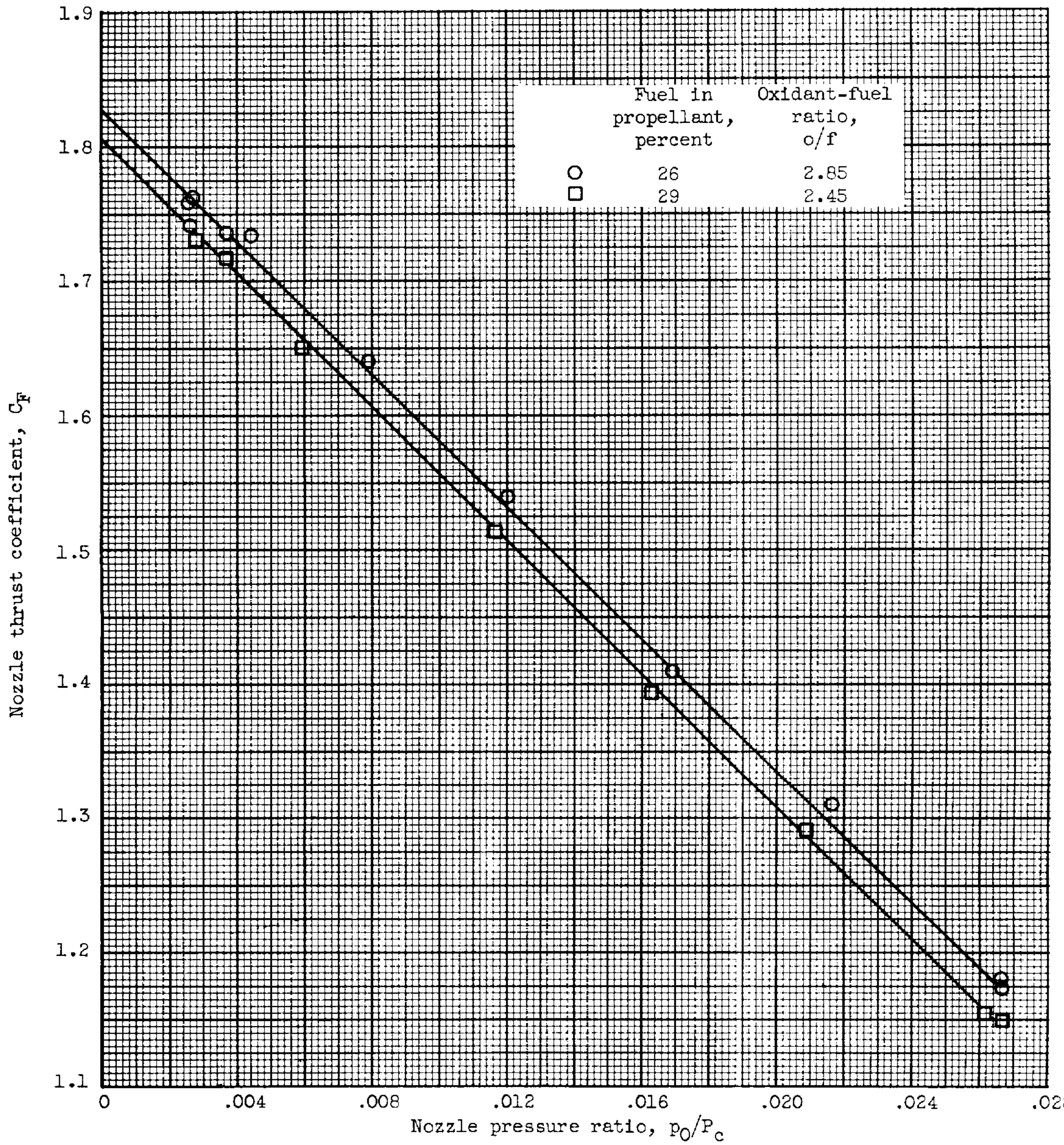
Figure 2. - Variation of nozzle performance parameters with nozzle pressure ratio for contoured nozzle with area ratio of 16.



(b) Specific impulse.

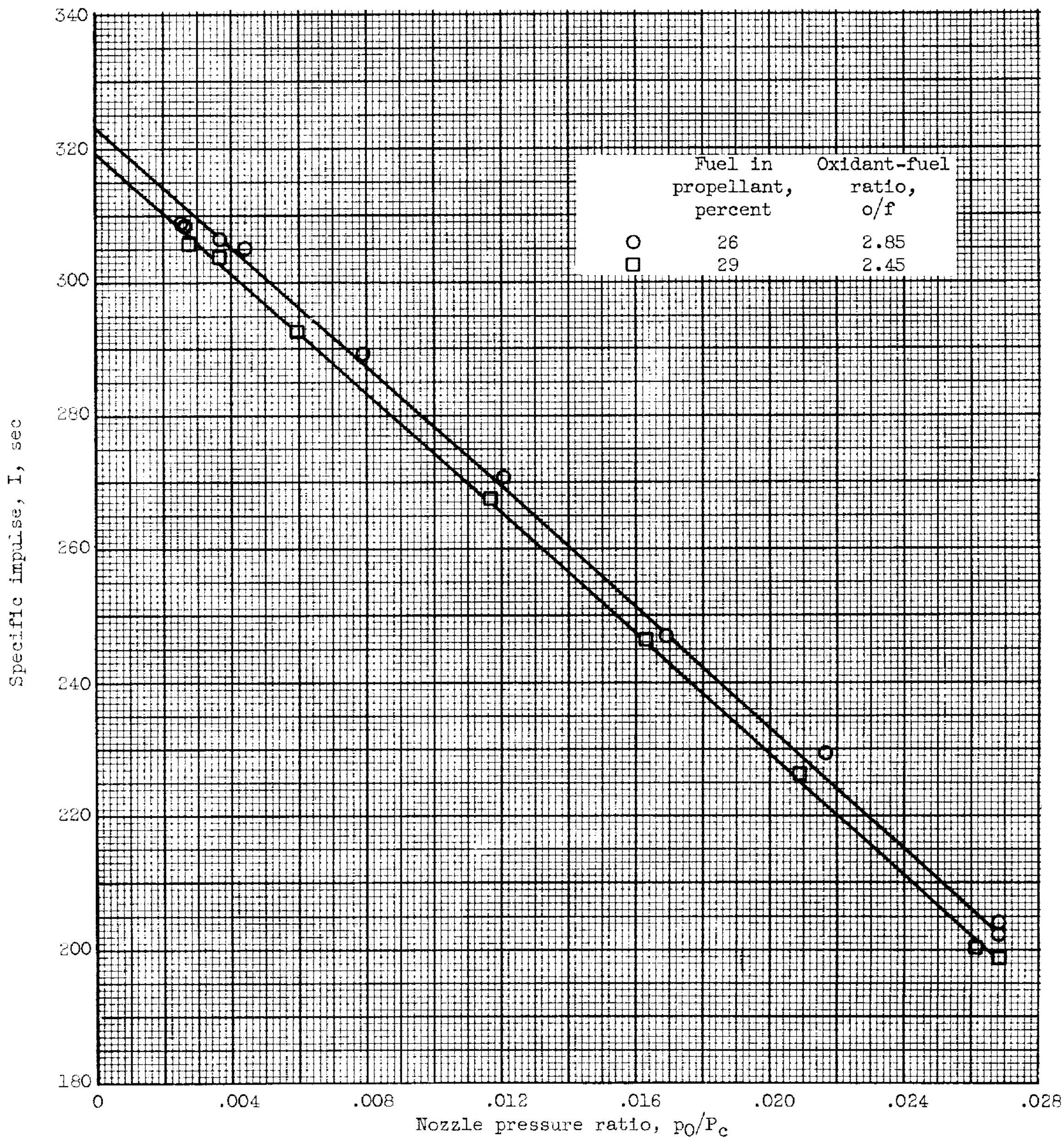
Figure 2. - Concluded. Variation of nozzle performance parameters with nozzle pressure ratio for contoured nozzle with area ratio of 16.

E-1403



(a) Nozzle thrust coefficient.

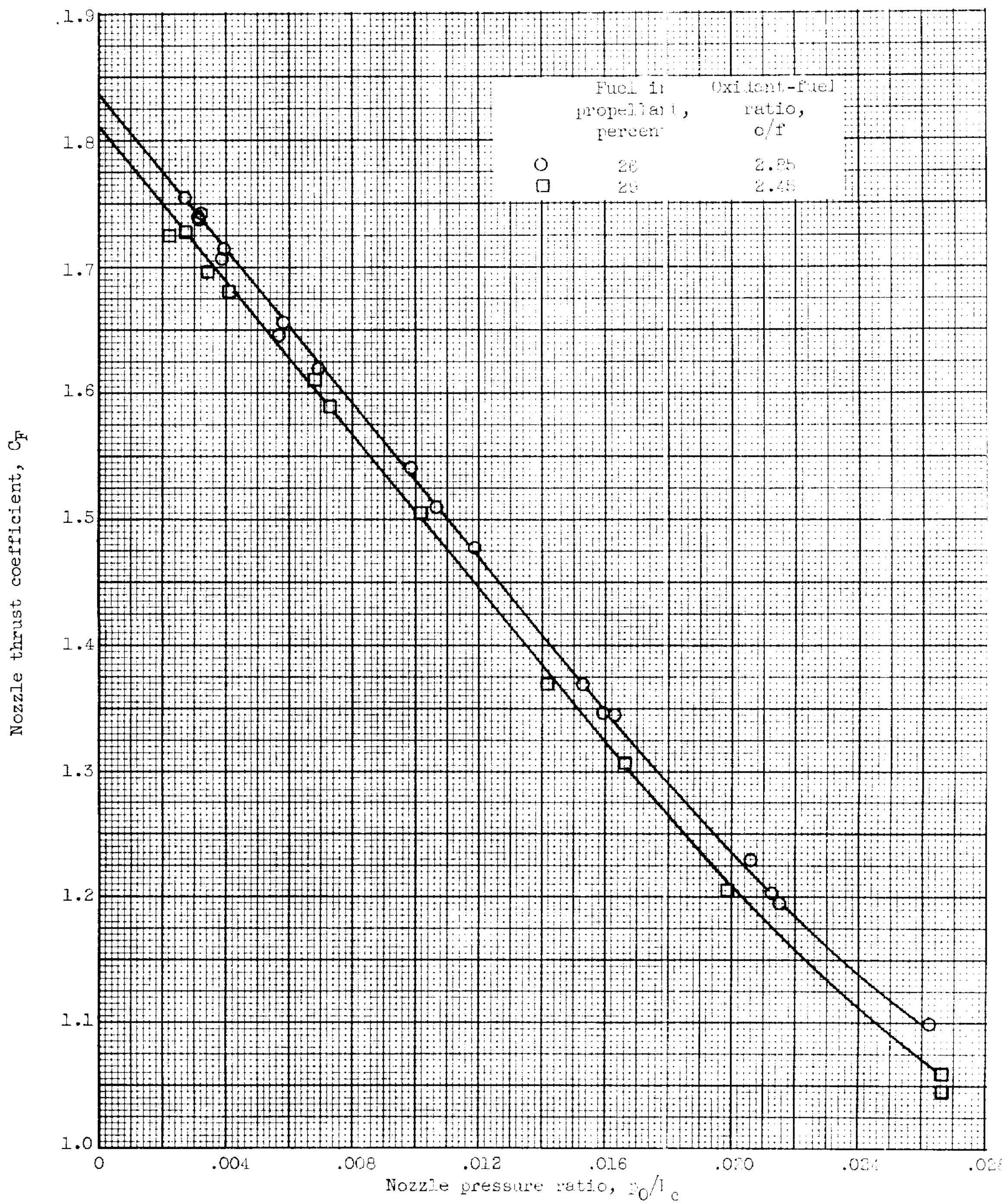
Figure 3. - Variation of nozzle performance parameters with nozzle pressure ratio for contoured nozzle with area ratio of 25.



(b) Specific impulse.

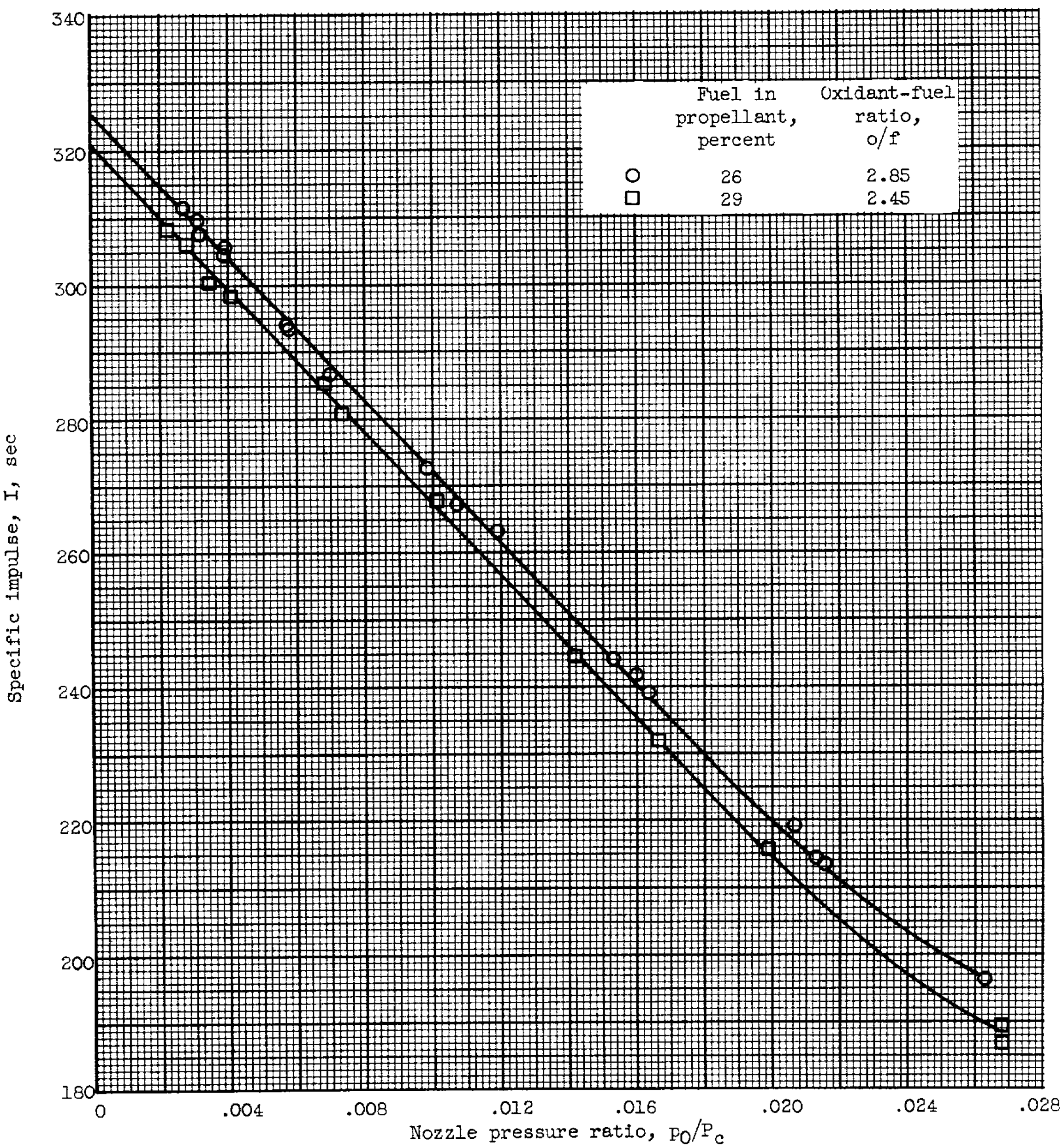
Figure 3. - Concluded. Variation of nozzle performance parameters with nozzle pressure ratio for contoured nozzle with area ratio of 25.

E-1403



(a) Nozzle thrust coefficient.

Figure 4. - Variation of nozzle performance parameters with nozzle pressure ratio for contoured nozzle with area ratio of 30.



(b) Specific impulse.

Figure 4. - Concluded. Variation of nozzle performance parameters with nozzle pressure ratio for contoured nozzle with area ratio of 30.

E-1403

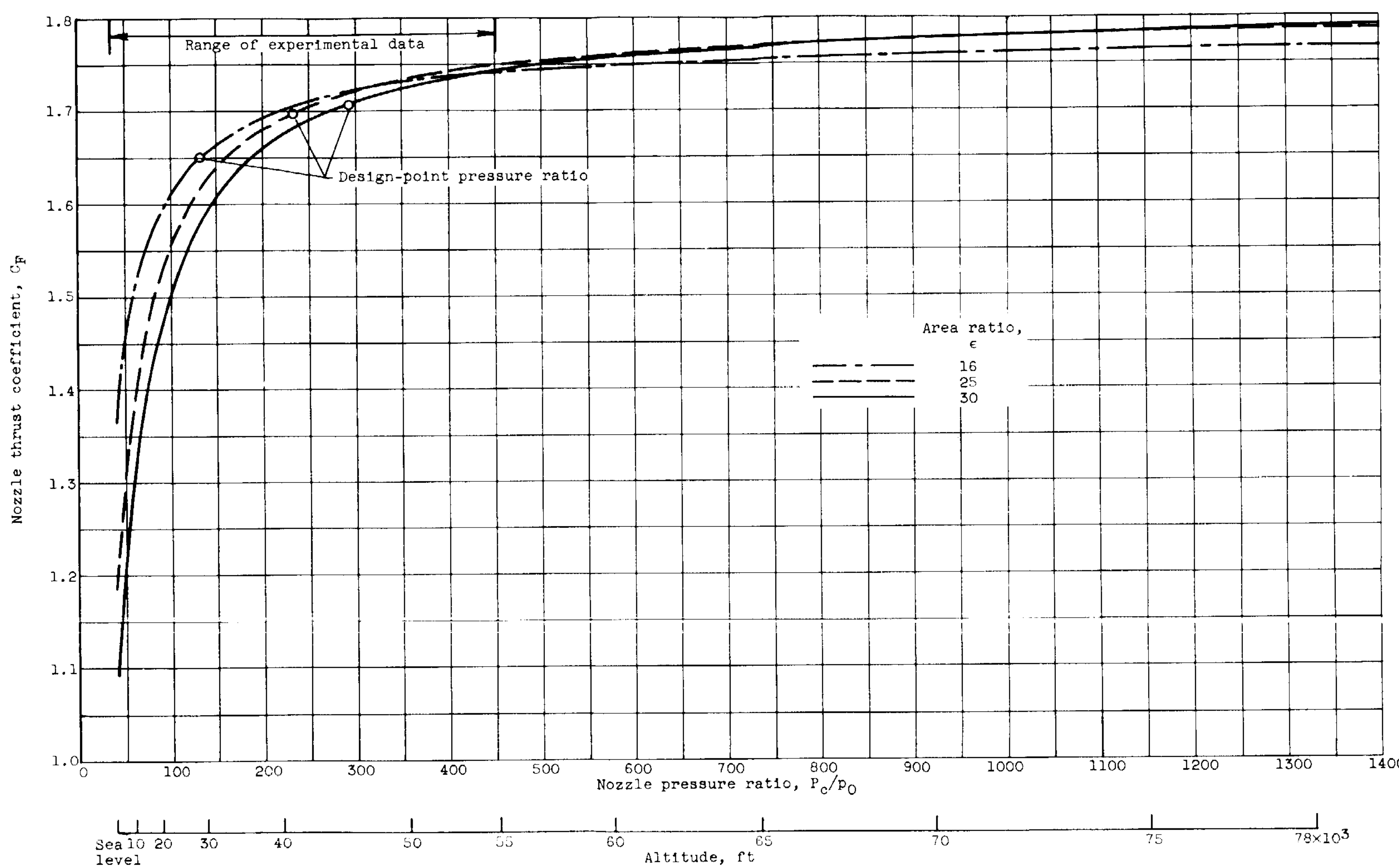


Figure 5. - Variation of thrust coefficient over range of pressure ratios for family of contoured nozzles with 29 percent fuel in propellant. Altitude scale based on thrust-chamber pressure of 600 pounds per square inch.

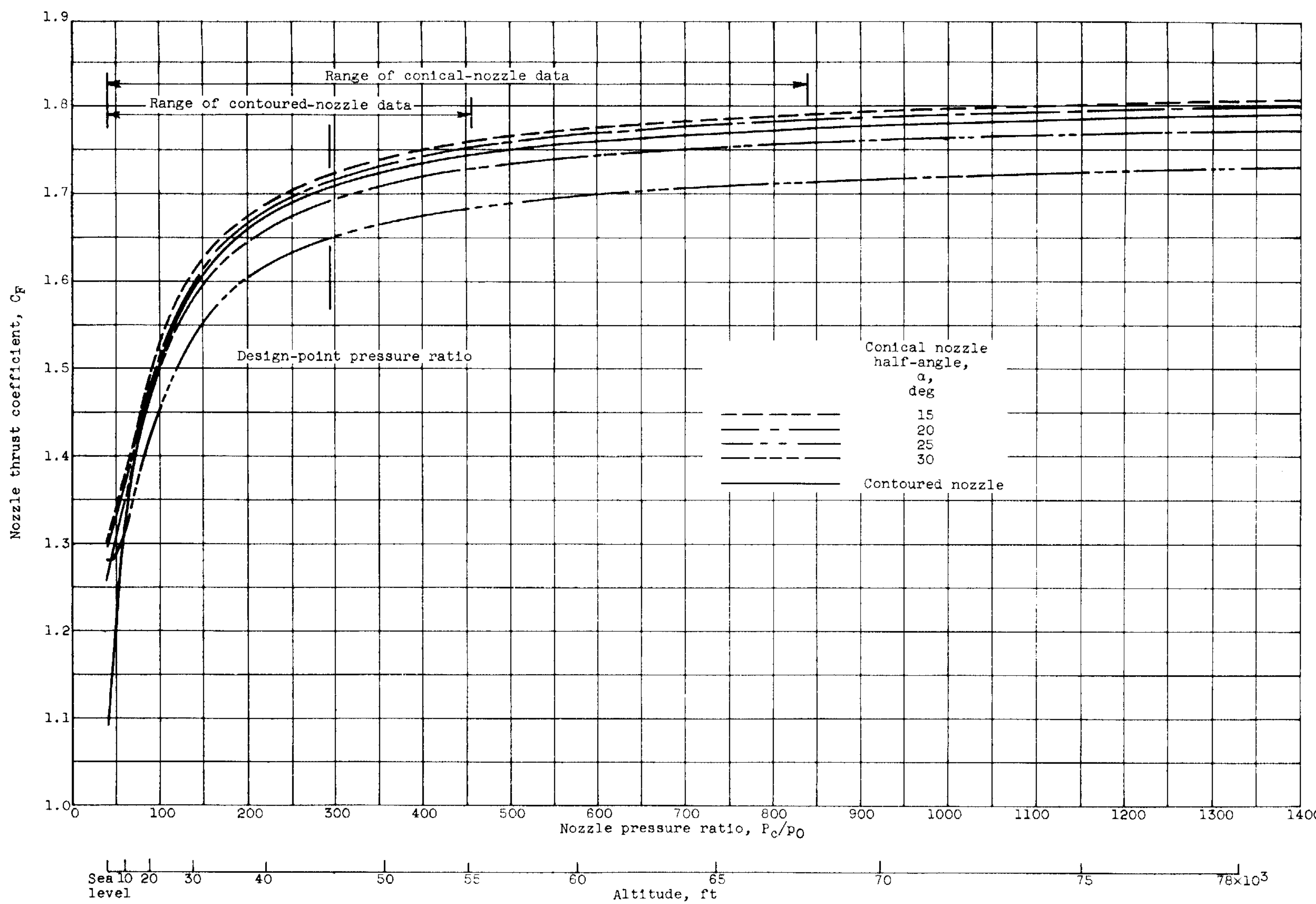
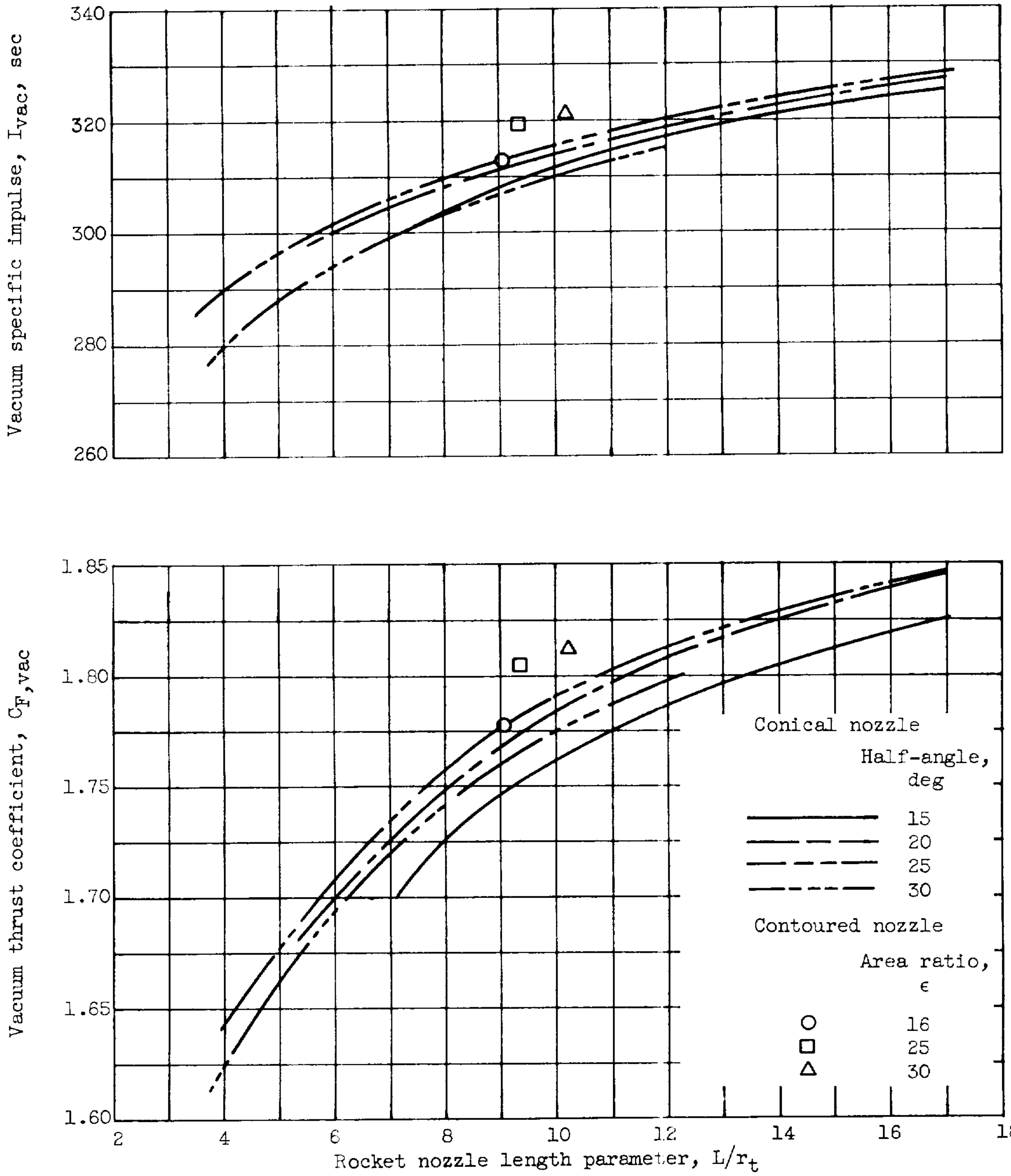


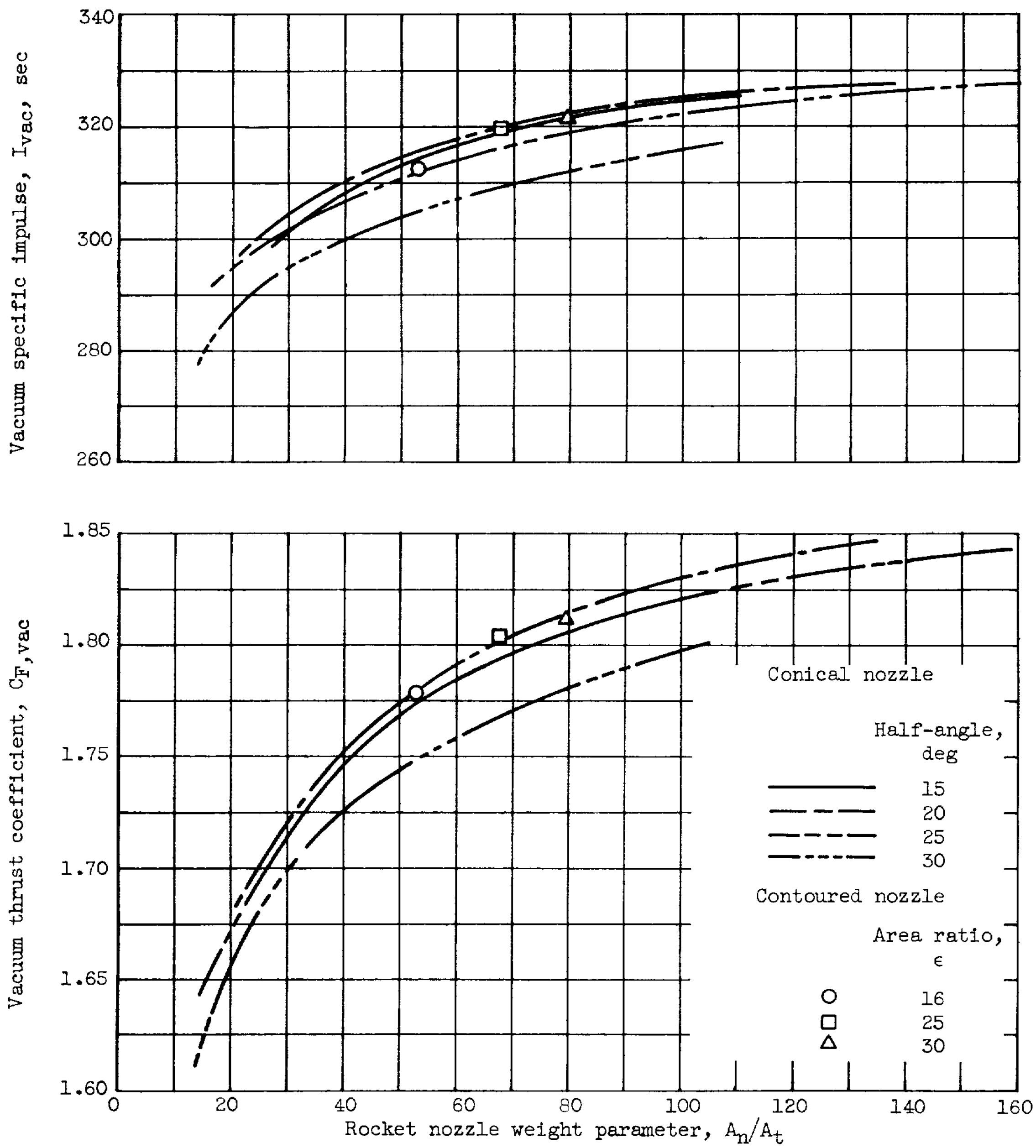
Figure 6. - Performance comparison of contoured nozzle with family of conical nozzles with nominal area ratio of 30 and 29 percent fuel in propellant. Altitude scale based on thrust-chamber pressure of 600 pounds per square inch.

E-1403



(a) Rocket nozzle length parameter.

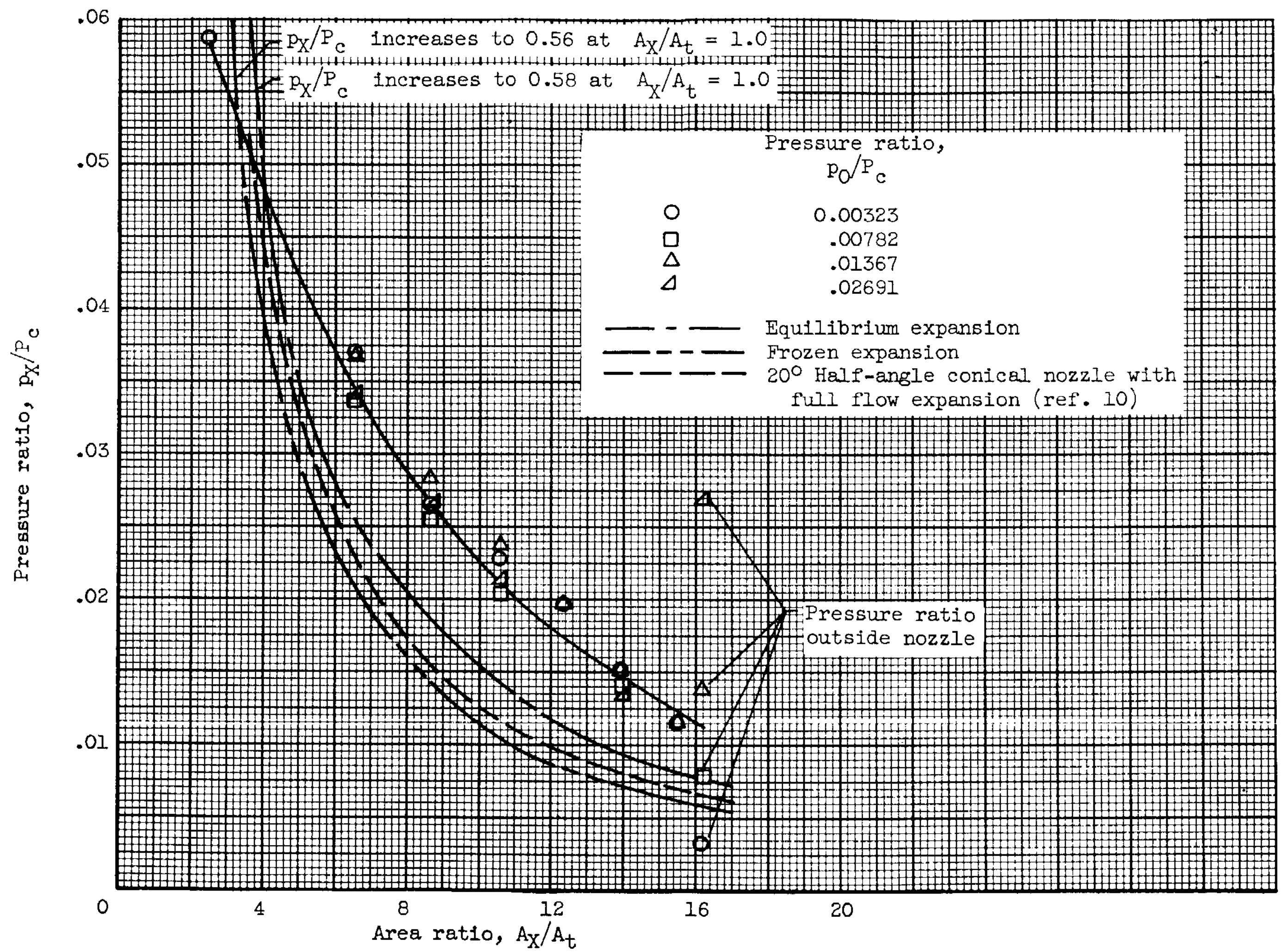
Figure 7. - Nozzle performance in vacuum with 29 percent fuel in propellant.



(b) Rocket nozzle weight parameter.

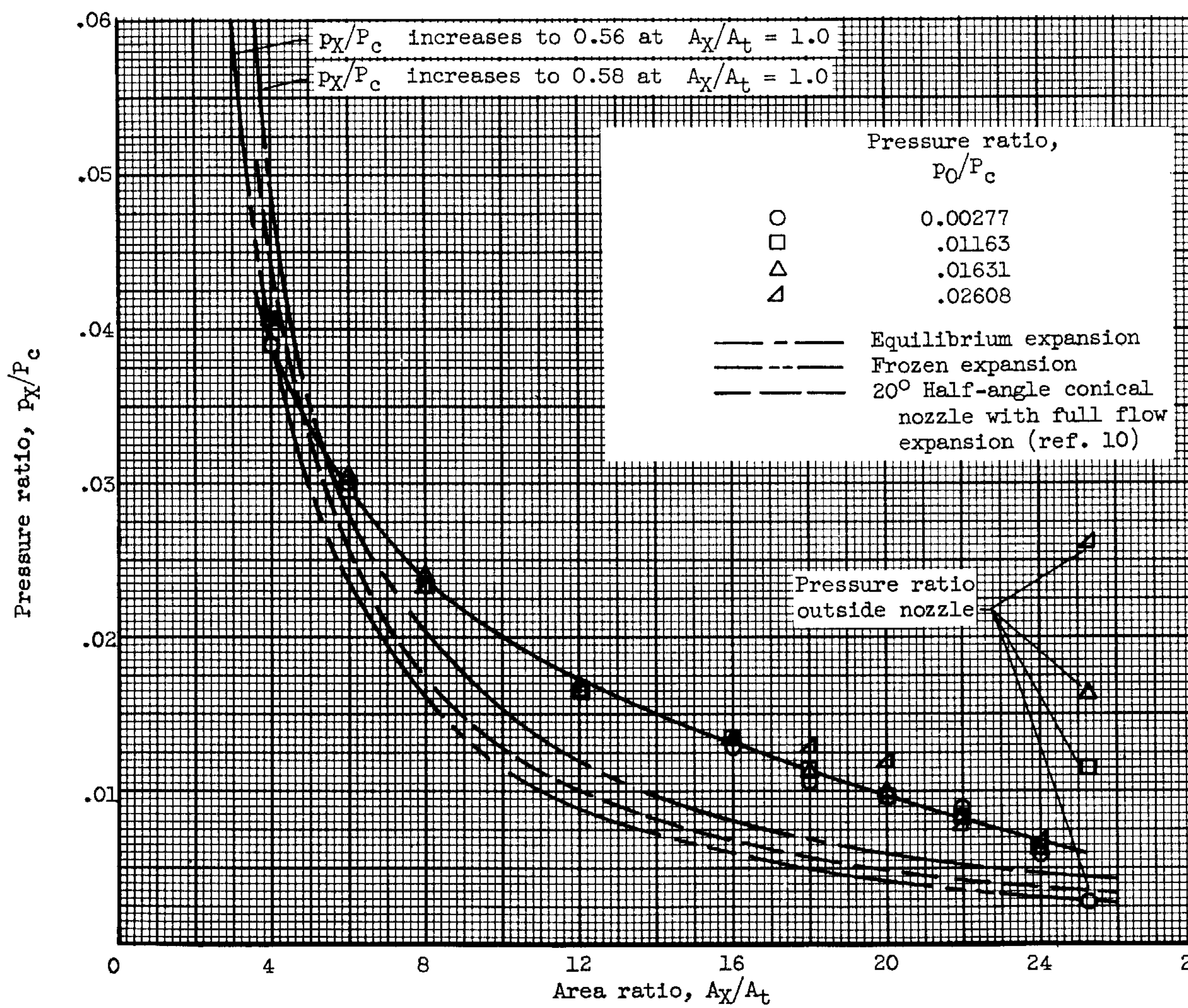
Figure 7. - Concluded. Nozzle performance in vacuum with 29 percent fuel in propellant.

E-1403



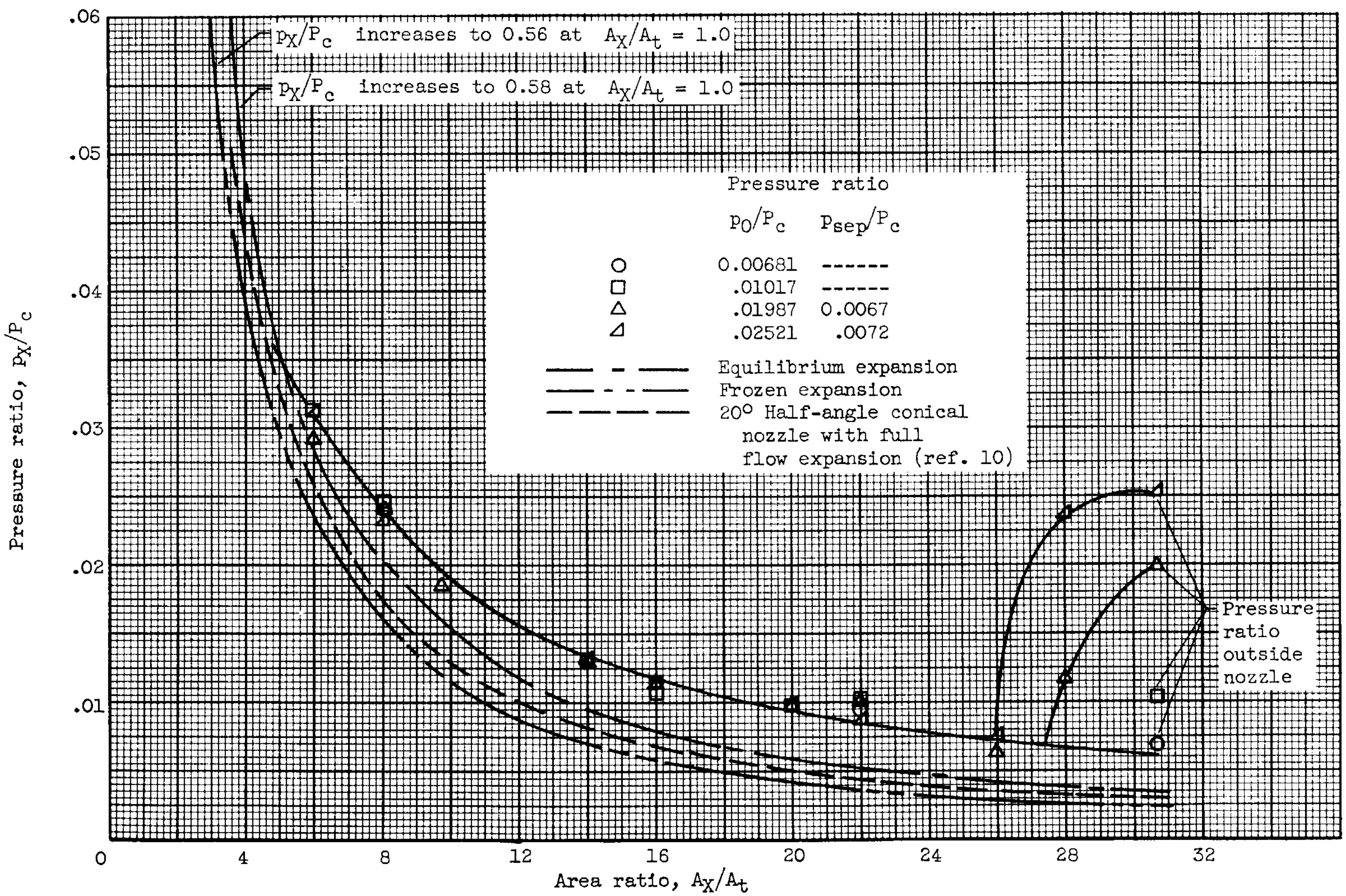
(a) Area ratio, 16.

Figure 8. - Nozzle pressure distribution for contoured nozzles with 29 percent fuel in propellant.



(b) Area ratio, 25.

Figure 8. - Continued. Nozzle pressure distribution for contoured nozzles with 29 percent fuel in propellant.



(c) Area ratio, 30.

Figure 8. - Concluded. Nozzle pressure distribution for contoured nozzles with 29 percent fuel in propellant.